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RESEARCH MEMORANDUM

PERFORMANCE OF A 20-INCH STEADY-FLOW RAM JET AT HIGH

ALTITUDES AND RAM-PRESSURE RATIOS

By Eugene Perchonok, William H. Sterbentz
and Fred A. Wilcox

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NATIONAL ADVISORY COMMITTEE
FOR AERONAUTICS

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SUMMARY

The results of an investigation conducted in the Cleveland altitude wind tunnel to determine the performance of a 20-inch ram jet are presented and discussed. The investigation was conducted at altitudes ranging from 7000 to 41,500 feet and at ram-pressure ratios equivalent to free-stream Mach numbers as great as 1.84 using preheated 62-octane fuel. Supplementary tests to determine any change in performance caused by changing the fuel to preheated 100-octane were also made. An extension of the methods of data reduction and of the generalizing performance parameters applicable at supersonic Mach numbers and over a wide range of operating conditions is presented. The magnitudes of the total-pressure losses across the various phases of the ram-jet cycle are analyzed and discussed.

At an equivalent free-stream Mach number of 1.84 and a gas total-temperature ratio across the engine of 5.7, the equivalent sea-level net thrust was 8135 pounds. For these conditions, the over-all efficiency was 12.6 percent and the combustion efficiency was 70.3 percent. The corresponding net-thrust coefficient was 0.74. The investigation also showed that no change in the performance or operating range of the engine occurred when the fuel was changed from preheated 62-octane to preheated 100-octane gasoline.

INTRODUCTION

Experiments have been conducted at the NACA Cleveland laboratory to determine the feasibility of operating a ram jet at high altitudes and at ram-pressure ratios equivalent to supersonic flight speeds. Performance studies of a ram jet at equivalent free-stream Mach numbers to 1.26 and at altitudes to 30,000 feet were made at this laboratory and are reported in references 1 and 2. Other ram-jet studies (references 3 to 6) present subsonic ram-jet performance.

In the present investigation made to extend the performance data,
dry refrigerated air was admitted to a 20-inch ram jet mounted in the
altitude wind tunnel. The desired ram-pressure ratio across the engine
was obtained by throttling the inlet air to the ram jet from approxi-
mately sea-level pressure and adjusting the static pressure in the
tunnel. The performance of the engine was studied at altitudes up to
41,500 feet and ram-pressure ratios equivalent to Mach numbers as great
as 1.84. The selection of the ram-jet configuration used in this study
was based on the results of previous investigations (references 1 and 2).

The results obtained in this investigation are summarized and the
effect of a variation in combustion-chamber length on engine performance
at high altitudes and at ram-pressure ratios equivalent to supersonic
flight speeds is discussed. The development of the parameters by means
of which the performance of ram jets can be generalized to any desired
operating conditions, originally presented in reference 1, has been
amplified to include additional variations encountered in the greater
operating range covered in this investigation. In the expression of
the values of ram-pressure ratio at which the engine was operated in
terms of equivalent free-stream Mach number, the effect of shock losses
that would occur at supersonic flight velocities are included.

APPARATUS AND PROCEDURE

The 20-inch ram jet used in the investigation was mounted in the
altitude-wind-tunnel test section below a 7-foot chord wing, which was
supported at the tips by the wind-tunnel balance frame (fig. 1). Dry
refrigerated air was supplied to the ram jet through a pipe from the
wind-tunnel make-up air duct. This air was available in the make-up
air duct at approximately sea-level pressure and was throttled to pro-
vide the desired total pressure at the diffuser inlet. The ram jet
exhausted directly into the wind tunnel, in which the pressure altitude
was varied to obtain different values of ram-pressure ratio across the
unit. Restraint of the model by the ram pipe was obviated by a sealed
slip joint inserted between the ram pipe and the diffuser inlet. The
tunnel balance system could then be used to measure the thrust.

The diffuser had an 8° included angle, a 14-inch-diameter inlet,
and a 20-inch-diameter exit. The engine was operated with replaceable
20-inch-diameter, 5- and 12-foot combustion-chamber sections to which a
converging nozzle 2 feet long with a 16.8-inch-diameter exit was
attached. The shell was cooled by circulating water through copper
tubing wrapped around the combustion chamber and exhaust nozzle.

Two different fuel injector systems (fig. 2) were used with no
apparent change in operation of the engine. One system consisted of

four 1/4-inch steel tubes having a total of 68 No. 70 holes drilled
along the tube lengths. These holes were equally spaced along the
four tubes and extended to within 2 inches of the diffuser wall.
The other system consisted of seven steel tubes having the same number
and size of holes equally spaced along the tubes. The total of 11 tubes
were equally spaced and arranged in an 80° V pattern; the open end of
the V was 5 inches downstream of the diffuser inlet. The fuel was
injected directly upstream.

A flame holder (fig. 3), consisting of a grid of horizontal
and vertical 30° V's 1 inch across the open end and with $2\frac{1}{2}$-inch
vertex spacing, was mounted at the combustion-chamber inlet with the
vertices of the V's facing upstream. A gas pilot to start combustion
was built into the flame holder and ignition for the pilot was pro-
vided by a modified aircraft spark plug. The cold static-pressure
drop across the flame holder was 2.9 times the dynamic pressure at
the combustion-chamber inlet.

The steam heat-exchanger fuel-preheating system described in
reference 2 was also used in this investigation. The fuel temperature
was maintained at 220° ±40° F by regulating the steam flow. An
unleaded 62-octane gasoline (AN-F-22) was used. Supplementary tests
were also made using a leaded 100-octane gasoline (AN-F-28).

During operation, local hot spots were observed on the combustion-
chamber shell under the copper cooling coils. An experimental section
(fig. 4), which eliminated all tendencies toward local overheating,
was inserted as part of the combustion chamber. The section was con-
structed by seam-welding an outer shell corrugated to form a helical
cooling-water path to a smooth, cylindrical inner shell.

Static pressures, total pressures, and indicated temperatures
were measured with a survey rake mounted at the diffuser inlet.
These pressures and temperatures were used to compute the air flow
through the engine and the velocities at the diffuser inlet and exit.
The fuel flow was determined with a rotameter. Fuel temperatures
and pressures were measured at the injector manifold.

Data were obtained at pressure altitudes ranging from 7000 to
41,500 feet. The fuel-air ratio was varied from approximately 0.040
to 0.067. Under choking conditions at the exhaust nozzle (jet Mach
number greater than 1), the maximum fuel-air ratio was limited to
0.051 because the peak delivery rate of the fuel pump had been
reached and the minimum fuel-air ratio was limited to approximately
0.042 because the fuel-injector pressure dropped below the fuel vapor
pressure at this point. The inlet-air temperature was maintained at
10° ±10° F.

SYMBOLS

The following symbols are used in this paper:

A	cross-sectional area, square feet
C_F	net-thrust coefficient, $\dfrac{2\,F_n}{\gamma_0 p_0 A_3 M_0{}^2}$
c_p	specific heat at constant pressure, Btu per pound per $^\circ F$
F_j	jet thrust, pounds
F_n	net thrust, pounds
f/a	fuel-air ratio
g	acceleration of gravity, feet per second per second
h	lower heating value of fuel, 19,000 Btu per pound
J	mechanical equivalent of heat, foot-pounds per Btu
M	Mach number
m_g	mass gas flow, slugs per second
P	total pressure, pounds per square foot absolute
p	static pressure, pounds per square foot absolute
R	gas constant, foot-pound per pound per $^\circ F$
T	total temperature, $^\circ R$
t	static temperature, $^\circ R$
V	velocity, feet per second
W_a	air flow, pounds per second
W_f	fuel flow, pounds per second
γ	ratio of specific heat at constant pressure to specific heat at constant volume

δ ratio of absolute tunnel ambient pressure to abso-
          lute static pressure at NACA standard atmospheric
          conditions at sea level, $p_0/2116$

θ_4 ratio of absolute total temperature at exhaust-
          nozzle exit to absolute static temperature at
          NACA standard atmospheric conditions at sea level,
          $T_4/519$

η over-all efficiency, percent

η_b combustion efficiency, percent

τ_1 ratio of absolute total temperature at exhaust-
          nozzle exit to absolute total temperature at dif-
          fuser inlet, T_4/T_1

τ_2 ratio of absolute total temperature at exhaust-
          nozzle exit to absolute total temperature at
          combustion-chamber inlet, T_4/T_2

Subscripts:

0   equivalent free-stream condition

1   station 1, subsonic diffuser inlet

2   station 2, diffuser exit and combustion-chamber
          inlet

3   station 3, combustion-chamber exit

4   station 4, exhaust-nozzle exit

j   exhaust-jet condition at ambient pressure
          $(p_j = p_0)$

Performance parameters:

F_j/δ jet thrust reduced to NACA standard atmospheric
          conditions at sea level, pounds

F_n/δ net thrust reduced to NACA standard atmospheric
          conditions at sea level, pounds

$M_2\sqrt{T_2}$ combustion-chamber-inlet Mach number parameter

$\eta(1 + \sqrt{\tau_1})$ over-all efficiency parameter, percent

$\dfrac{\eta}{\eta_b}(1 + \sqrt{\tau_1})$ ideal over-all efficiency parameter, percent

$\dfrac{W_a}{\delta}\sqrt{\theta_4}$ air-flow parameter, pounds per second

$\dfrac{W_f \eta_b\, 3600}{\delta \sqrt{\theta_4}}$ fuel-consumption parameter, pounds per hour

$\dfrac{550\, W_f\, 3600}{F_n V_0}$ net-power specific fuel consumption, pounds per net thrust horsepower-hour

$\dfrac{550\, W_f\, 3600}{F_n V_0(1 + \sqrt{\tau_1})}$ net-power specific fuel consumption parameter, pounds per net thrust horsepower-hour

$\dfrac{550\, W_f\, 3600}{F_n V_0(1 + \sqrt{\tau_1})}\eta_b$ ideal net-power specific fuel consumption parameter, pounds per net thrust horsepower-hour

RESULTS AND DISCUSSION

At all fuel-air ratios at which the engine was operated, flame completely filled the combustion chamber. High-speed motion-picture studies indicated, however, that the burning, which appeared to be smooth but was accompanied by a steady buzzing sound, was actually of a high-frequency pulsating nature.

Below choking (jet Mach number less than 1), the flame emitted from the exhaust nozzle was continuous except for occasional flashes curling outward. Above choking (jet Mach number greater than 1), the operation of the engine seemed independent of tunnel ambient pressure and shock bands were clearly visible in the exposed flame. These shock bands were spaced approximately uniformly along the jet from the nozzle exit. Little flame color could be seen between the nozzle exit and the first bright band (fig. 5). (For an exposition of the aerodynamic characteristics of supersonic gas jets, see reference 7.)

In general, the data for this investigation have been reduced
and correlated by methods discussed in reference 1. In certain
instances, however, the performance parameters of reference 1 were
oversimplified because of the narrow operating range covered and
proved inadequate for correlating the data at the high Mach numbers
obtained in this investigation. A refinement and an extension of
the performance parameters and methods of data reduction applicable
to performance at high Mach numbers and over a wide operating range
are presented in the appendix.

The equivalent free-stream Mach number M_0 is taken as the
independent variable in presenting the data. This quantity is calcu-
lated from the equivalent free-stream total pressure and the tunnel
ambient static pressure. For values of M_0 less than 1, the diffuser-
inlet total pressure was taken as the equivalent free-stream total
pressure. For values of M_0 greater than 1, supersonic diffuser
losses were added to the measured subsonic diffuser-inlet total pres-
sure to obtain the equivalent free-stream total pressure. The
assumptions used in obtaining the equivalent free-stream total pres-
sure and M_0 from the measured diffuser-inlet total pressure and the
tunnel ambient static pressure for $M_0 > 1$ are given in the appendix.

The relations of the performance parameters to M_0 for the
5-foot combustion-chamber engine are presented in figures 6 to 20.
An investigation using 100-octane gasoline (AN-F-28) showed no change
in the performance of the engine as compared with the performance when
62-octane gasoline (AN-F-22) was used. For this reason, no data for
the performance study using 100-octane gasoline are presented. With
the exception of the combustion-efficiency data, the test points for
the 12-foot combustion chamber fell along the curves established for
the 5-foot combustion chamber (figs. 6 to 19). Therefore, only the
jet-thrust (figs. 6 and 7) and the combustion-efficiency data (fig. 21)
for the 12-foot combustion chamber are presented.

The highest equivalent free-stream Mach number for operation
with the 5-foot combustion chamber was 1.84. This maximum was set
not by an operational limit of the engine but by the pumping capacity
of the test apparatus. With the 12-foot combustion chamber, the
highest M_0 at which the engine could be operated before blow-out
was 1.19.

The maximum jet thrust developed by the engine with the 5-foot
combustion chamber, reduced to sea-level conditions F_j/δ (fig. 6)
was 14,690 pounds at $M_0 = 1.84$. The actual jet thrusts F_j meas-
ured and the altitudes at which these data were obtained are presented
in figure 7. The pressure-altitude contours are based on the reduced
jet-thrust curve of figure 6.

the engine τ_1 of 5.7, and a combustion efficiency η_b of 70.3 percent, the over-all efficiency η was 12.6 percent (fig. 14). The corresponding net-power specific fuel consumption $\dfrac{550\ W_f\ 3600}{F_n V_0}$ was 1.06 pounds per net thrust horsepower-hour (fig. 15).

The variation of the ultimate exhaust-jet Mach number M_j with M_0 is shown in figure 16. Choking at the exhaust nozzle begins approximately at $M_0 = 1.16$. Also presented in figure 16 are theoretical curves of M_j as a function of M_0 and total-pressure ratio across the engine P_j'/P_0, assuming the ratio of specific heats at the free-stream condition γ_0 and the exhaust-jet condition γ_j equal to 1.4 and 1.3, respectively. These curves are based on equation (13) of the appendix. The correlation of these theoretical curves and the experimental data indicates the magnitude of the total-pressure loss through the engine.

A discussion and evaluation of the various types of pressure loss in a ram jet are given in references 8 and 9. From the figures of reference 8, the magnitude of the total-pressure losses across the various phases of the ram-jet cycle can be estimated. At a typical engine-performance condition of $M_0 = 1.70$, $\tau_1 = 6.0$, and $M_2 = 0.14$, the theoretical total-pressure ratio is 0.93 across the normal shock in a convergent-divergent supersonic diffuser with optimum contraction ratio (reference 8, fig. 2), 0.96 across the flame holder with a cold pressure-drop coefficient of 2.9 (reference 8, fig. 5), and 0.91 across a constant-area combustion chamber as a result of burning (reference 8, fig. 6). Inasmuch as the total-pressure ratio across the engine was 0.75 at $M_0 = 1.70$ (fig. 16), there remains a total-pressure ratio of 0.92 attributable to losses resulting from shell friction, fuel-injector drag, and subsonic diffuser and exhaust-nozzle inefficiency.

It is impossible to reduce the greatest of the total-pressure losses, the loss caused by combustion in a constant-area tube, if maximum thrust coefficients are desired because the ram jet must be operated at high values of τ_1 and M_2. With continued research and development, it should be possible to decrease the total-pressure losses caused by the flame holder and to decrease the fuel-injector drag, the shell friction, and the diffuser and exhaust-nozzle inefficiencies.

The data in figure 16 can be used to extrapolate the performance of the engine to Mach numbers in excess of those at which the engine

was operated because all of the engine losses, except the supersonic diffuser losses, will remain relatively constant for conditions at which $M_j > 1$ (τ_1 and M_2 constant). The decrease in the theoretical total-pressure recovery across the supersonic portion of a converging-diverging diffuser of optimum contraction ratio with an increase in M_0 can be determined from figure 2 of reference 8. By application of these calculated losses to the total-pressure ratio across the engine at $M_j = 1$ (fig. 16), an extrapolated curve of M_j can be obtained from an extension of the pressure-ratio contours on that figure.

If the absolute total-temperature ratio τ_2 and M_0 are assumed for some desired operating condition, then the Mach number at the combustion-chamber entrance M_2 can be estimated from figure 17. In this figure, the combustion-chamber-inlet Mach number parameter $M_2\sqrt{\tau_2}$ was practically independent of M_0 at Mach numbers greater than 1, although choking first occurred at $M_0 = 1.16$.

The air-flow parameter $\dfrac{W_a}{\delta}\sqrt{\theta_4}$ and the fuel-consumption parameter

$\dfrac{W_f \eta_b \, 3600}{\delta \sqrt{\theta_4}}$ are plotted as functions of M_0 in figures 18 and 19, respec-

tively. The air-flow and fuel-consumption parameters were not affected by choking at the exhaust nozzle and continued to increase at $M_0 > 1.16$. The fuel-consumption parameter is presented as a function of τ_2 in

addition to M_0 to include the effect of the term $\left(1 - \dfrac{\tau_2}{T_4}\right)$ of equation (28), reference 1.

Combustion-efficiency data are presented as a function of fuel-air ratio f/a for the 5-foot and 12-foot combustion chambers in figures 20 and 21, respectively. A comparison of these data indicates that lengthening the combustion chamber from 5 to 12 feet improved the combustion efficiency about 10 percent over the fuel-air-ratio range at which the engine was operated. As in reference 2, the heat losses through the ram-jet shell were not included in the calculation of combustion efficiency. If these heat losses were included, the combustion-efficiency values would be approximately 3 percent higher for the engine having the 12-foot combustion chamber and 1 percent higher for the engine with the 5-foot combustion chamber. The numbers opposite each point indicate the values of combustion-chamber-inlet static pressure p_2, combustion-chamber-inlet Mach number M_2, and exhaust-nozzle-outlet static pressure p_0 (ambient static pressure). These variables are the ones thought to influence combustion efficiency if the combustion-chamber-inlet temperature T_2 is constant.

It was difficult to separate quantitatively the effect of each variable on combustion efficiency. As mentioned in references 1 and 2, a decrease in the combustion-chamber static-pressure level or an increase in M_2 usually resulted in a decrease in combustion efficiency. With approximately static sea-level pressure maintained at the combustion-chamber inlet, pressure altitude apparently had no effect on η_b after choking conditions were attained at the exhaust nozzle. No investigation was made of the effect of reduced p_2 on combustion efficiency and blow-out.

The greatest combustion-chamber-inlet velocity V_2 at which the burner was operated with a 5-foot combustion chamber was 152 feet per second. This velocity was measured at a pressure altitude of 27,000 feet when the engine was operating under choking conditions at the exhaust nozzle $(M_0 = 1.33)$ and at a total-temperature ratio across the combustion chamber $\tau_2 = 5.0.$. Blow-out did not occur at this condition and it should not be considered as the limiting combustion-chamber-inlet velocity for this burner. The maximum V_2 attained with a 12-foot combustion chamber was 140 feet per second at a pressure altitude of 23,000 feet. This value was obtained when the engine was operating under choking conditions at the exhaust nozzle $(M_0 = 1.19)$.

SUMMARY OF RESULTS

From an investigation of the performance of a 20-inch ram jet over a wide range of pressure altitudes and equivalent free-stream Mach numbers, the following results were observed:

1. A net thrust of 8135 pounds reduced to standard sea-level conditions, a net-thrust coefficient of 0.74, and an over-all efficiency of 12.6 percent were attained at the maximum equivalent free-stream Mach number of 1.84 at which the engine with a 5-foot combustion chamber was operated. The corresponding specific fuel consumption was 1.06 pounds per net thrust horsepower-hour. At this condition the total-temperature ratio across the engine was 5.7 and the combustion efficiency was 70.3 percent.

2. The engine with a 5-foot combustion chamber was operated at combustion-chamber-inlet velocities up to 152 feet per second. This velocity was attained during choking at the exhaust nozzle and at a ratio of absolute total temperature at the exhaust-nozzle exit to that at the combustion-chamber inlet of 5.0.

3. In the range investigated the principal total-pressure loss was caused by combustion. At an equivalent free-stream Mach number of 1.70, a combustion-chamber-inlet Mach number of 0.14, and at a total-temperature ratio across the engine of 6.0, the total-pressure ratio across the engine was 0.75. For these conditions the estimated total-pressure ratios across the various phases of the ram-jet cycle that contribute to total-pressure losses were as follows:

Total-pressure ratio across the supersonic portion
  of diffuser 0.93
Total-pressure ratio across flame holder 0.96
Total-pressure ratio across combustion chamber as
  a result of combustion 0.91
Total-pressure ratio caused by losses across sub-
  sonic portion of diffuser, exhaust nozzle, fuel
  injector, and combustion chamber as a result of
  shell friction 0.92

4. At a constant gas total-temperature ratio across the engine and a constant combustion efficiency, the performance curves of the engine were not noticeably affected by changes in the combustion-chamber length. Although increasing the combustion-chamber length from 5 to 12 feet improved the combustion efficiency, the operating range of the engine was reduced.

5. No change in the performance or operating range of the engine was observed when the fuel was changed from preheated unleaded 62-octane to preheated leaded 100-octane gasoline.

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APPENDIX -- METHODS OF DATA REDUCTION

The data presented in this paper have been reduced and correlated for the most part by the methods discussed in reference 1. Because some of the data were obtained at equivalent free-stream Mach numbers M_0 markedly greater than those in reference 1, the methods of data reduction have been modified to include the effects of variables that are negligible at the lower equivalent free-stream Mach numbers.

Equivalent Free-Stream Mach Number

The equivalent free-stream Mach number M_0 was determined from the relation

$$M_0^2 = \frac{2}{\gamma_0 - 1} \left[\left(\frac{P_0}{P_0} \right)^{\frac{\gamma_0 - 1}{\gamma_0}} - 1 \right] \tag{1}$$

For values of M_0 less than 1, the diffuser-inlet total pressure P_1 was taken as the free-stream total pressure P_0.

For values of M_0 greater than 1, the measured inlet total pressure was adjusted in the ratio of the total pressures across an assumed supersonic diffuser. This pressure ratio was taken as the theoretical pressure ratio across a normal shock at the throat of a convergent-divergent supersonic diffuser designed to allow shock entrance at the pertinent Mach number. (See reference 10.) Because even more favorable pressure ratios have been obtained on a different type of supersonic diffuser (references 11 to 13), this assumption seemed reasonable.

Exhaust-Nozzle-Exit Temperature

As derived in reference 1, the exhaust-nozzle-exit temperature T_4 was determined from the expression

$$T_4 = \frac{p_4 A_4 F_j}{gRm_g^2} - \frac{p_4 A_4^2 (p_4 - p_0)}{gRm_g^2} + \frac{\left[F_j - A_4 (p_4 - p_0) \right]^2}{2gJc_{p,4}m_g^2} \tag{2}$$

When the ultimate exhaust-jet Mach number M_j is less than 1, an error in T_4 of less than 2 percent results if p_4 is assumed equal to p_0.

This assumption, however, leads to increasingly erroneous values of T_4 for conditions of M_j greater than 1. It is necessary to determine p_4 more accurately at the conditions for which M_j is greater than 1.

Because M_4 is approximately equal to 1 for all conditions of M_j greater than 1

$$\frac{P_4}{p_4} = \left(\frac{\gamma_4 + 1}{2}\right)^{\frac{\gamma_4}{\gamma_4 - 1}} \tag{3}$$

Further, the jet thrust F_j can be expressed as

$$F_j = \frac{2\gamma_4}{\gamma_4 - 1} p_4 A_4 \left[\left(\frac{P_4}{p_4}\right)^{\frac{\gamma_4 - 1}{\gamma_4}} - 1\right] + A_4(p_4 - p_0) \tag{4}$$

From a combination of equations (3) and (4), the following expression results:

$$p_4 = \frac{F_j}{A_4(\gamma_4 + 1)} + \frac{p_0}{(\gamma_4 + 1)} \tag{5}$$

Equation (5) was used to determine p_4 for all data presented for conditions where M_j was greater than 1.

Over-All Efficiency Parameter

The over-all efficiency parameter was modified to include the effect of variations in the total-temperature ratio across the engine τ_1 (τ_2 assumed equal to τ_1) on the over-all efficiency. This effect became noticeable at high values of M_0. The over-all efficiency of the engine is

$$\eta = \frac{\text{output}}{\text{input}} = \frac{F_n V_0}{W_f h J} \tag{6}$$

It is assumed that the ratio of specific heats γ does not change through the ram jet, that no energy loss occurs in the diffusion and effusion processes, and that the gas flow is essentially the air flow. Then the net thrust F_n, free-stream velocity V_0, and fuel consumption W_f can be stated as follows:

$$F_n = \frac{W_a}{g} V_0 \left(\sqrt{\tau_1} - 1 \right) \tag{7}$$

$$V_0 = \frac{M_0 \sqrt{\gamma_0 gRT_1}}{\sqrt{1 + \frac{\gamma_0 - 1}{2} M_0^2}} \tag{8}$$

$$W_f = \frac{W_a c_{p,0} (T_4 - T_1)}{h \eta_b} \tag{9}$$

Substitution of equations (7), (8), and (9) into equation (6) gives, as an approximation, the over-all efficiency in terms of M_0, η_b, and τ_1. Thus

$$\eta = \frac{(\gamma_0 - 1) M_0^2}{1 + \frac{\gamma_0 - 1}{2} M_0^2} \left(\frac{\eta_b}{1 + \sqrt{\tau_1}} \right) \tag{10}$$

Equation (10) shows that the quantity of $\eta(1 + \sqrt{\tau_1})$ is primarily a function of M_0 and η_b. The quantity $\eta(1 + \sqrt{\tau_1})$ was therefore plotted as a function of M_0 and η_b.

Similarly, by substitution of equation (10) into equation (6), the net-power specific fuel consumption parameter $\dfrac{550 \, W_f \, 3600}{F_n V_0 (1 + \sqrt{\tau_1})}$ is primarily also a function of M_0 and η_b. This quantity was therefore plotted as a function of M_0 and η_b.

Ultimate Exhaust-Jet Mach Number

For conditions at which the ultimate exhaust-jet Mach number M_j was less than 1, its value was calculated by the method described in reference 1. For values greater than 1, M_j was calculated in the following manner:

Because there is relatively small loss in total pressure between stations 4 and j, jet Mach number M_j can be expressed as

$$M_j{}^2 = \frac{2}{\gamma_j - 1}\left[\left(\frac{P_4}{P_0}\right)^{\frac{\gamma_j-1}{\gamma_j}} - 1\right] \qquad (11)$$

Inasmuch as M_4 is approximately equal to 1.0 for all conditions of M_j greater than 1, P_4 from equation (3) can be substituted into equation (11). Thus, assuming γ_j equal to γ_4

$$M_j{}^2 = \frac{2}{\gamma_j - 1}\left[\frac{\gamma_j + 1}{2}\left(\frac{P_4}{P_0}\right)^{\frac{\gamma_j-1}{\gamma_j}} - 1\right] \qquad (12)$$

The exhaust-nozzle-exit static pressure p_4 is given by equation (5).

Total-Pressure Ratio Across Engine

The theoretical curves of M_j as a function of M_0 and the total-pressure ratio across the engine P_j/P_0 presented in figure 16 were calculated by means of the following expression:

$$\frac{P_j}{P_0} = \frac{\left[1 + \frac{\gamma_j - 1}{2} M_j{}^2\right]^{\frac{\gamma_j}{\gamma_j-1}}}{\left[1 + \frac{\gamma_0 - 1}{2} M_0{}^2\right]^{\frac{\gamma_0}{\gamma_0-1}}} \qquad (13)$$

Equation (13) is based on the assumption that p_j is equal to p_0.

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Figure 1.-Installation of 20-inch ram jet with 12-foot combustion chamber and 16.8-inch-diameter exhaust nozzle in altitude wind tunnel.
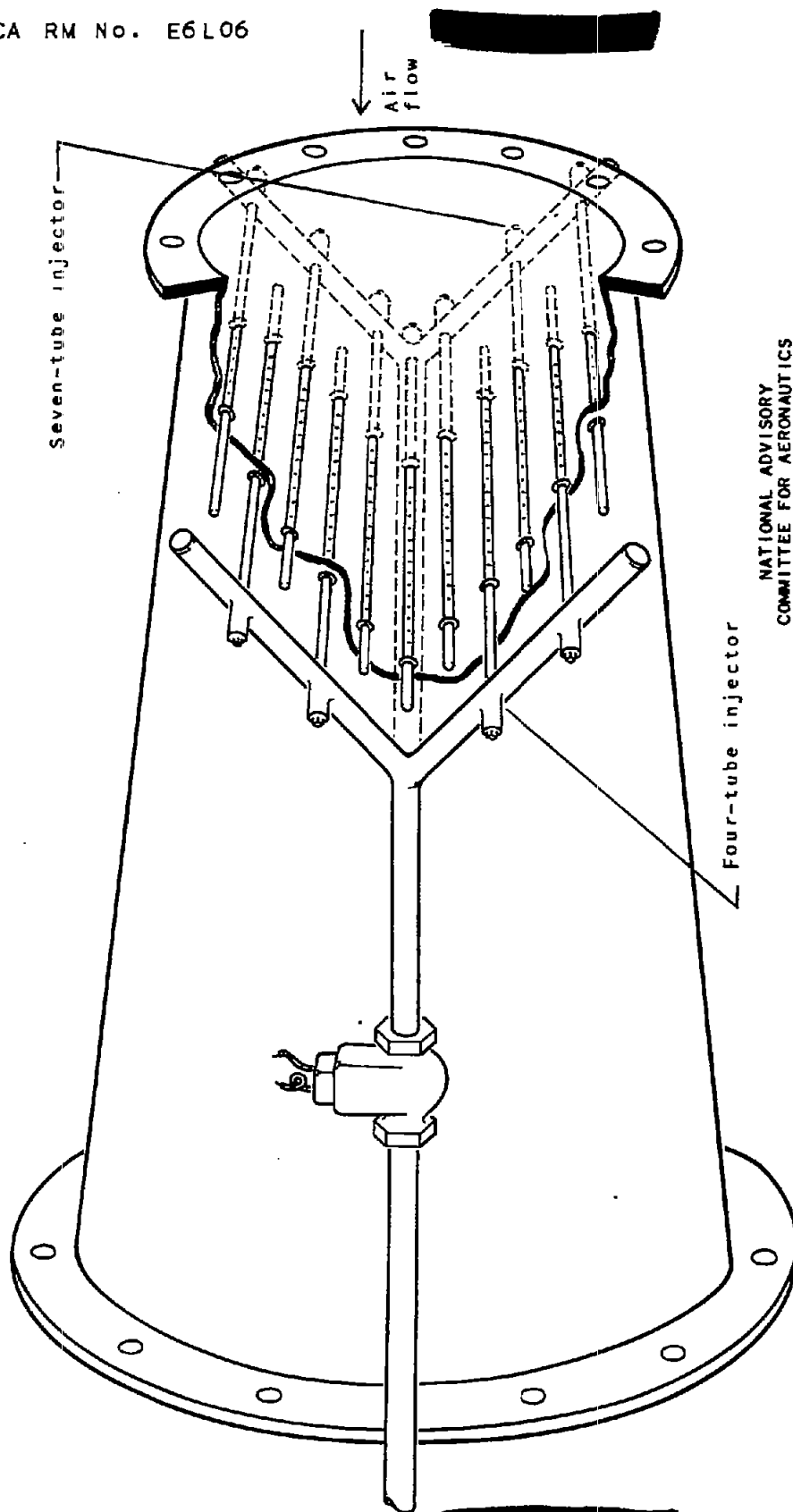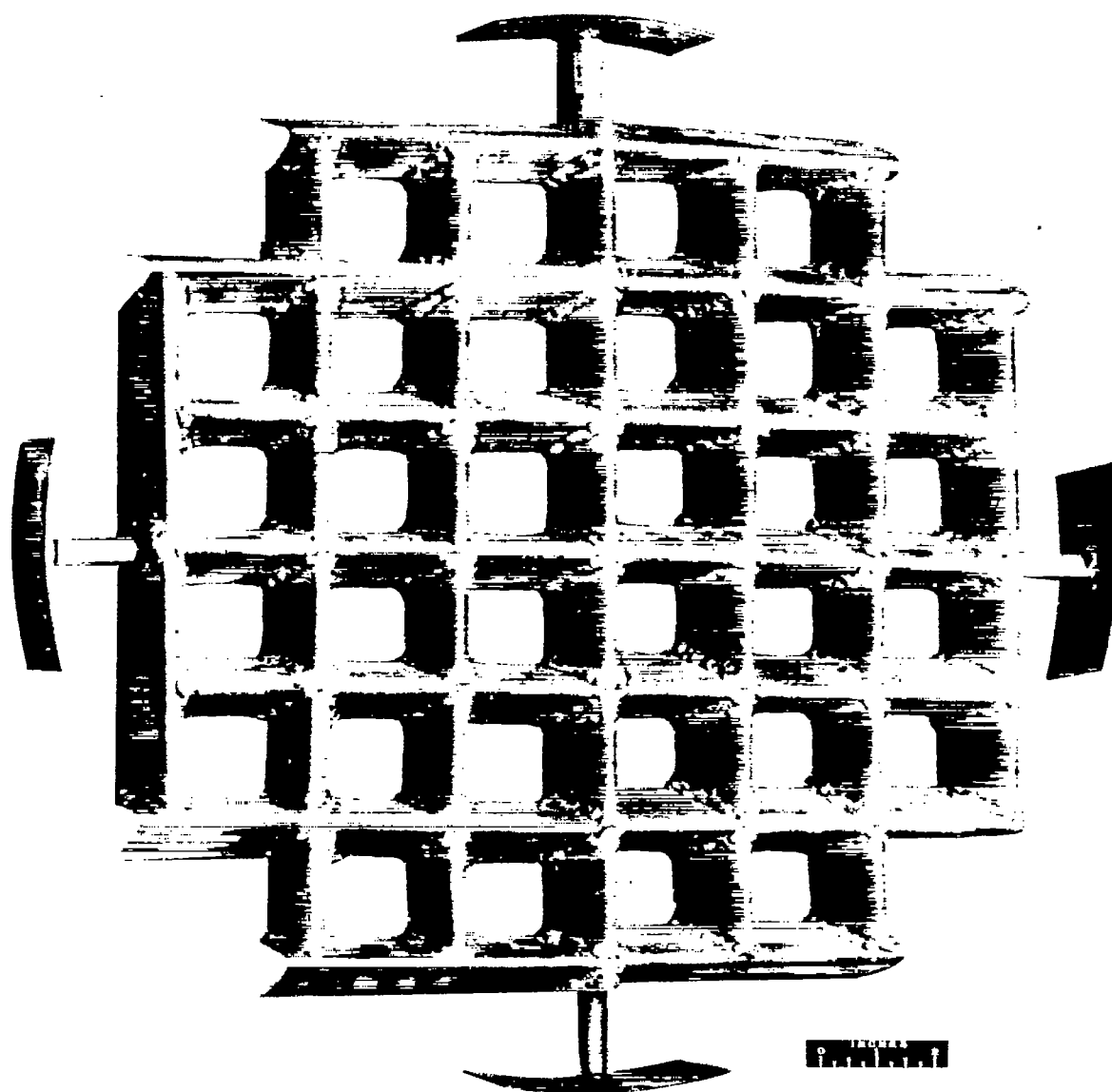
684

NATIONAL ADVISORY
COMMITTEE FOR AERONAUTICS

Figure 2. - Installation of four- and seven-tube fuel injectors in 20-inch ram-jet diffuser.

684

NACA
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Figure 3. - Flame holder used in wind-tunnel investigation
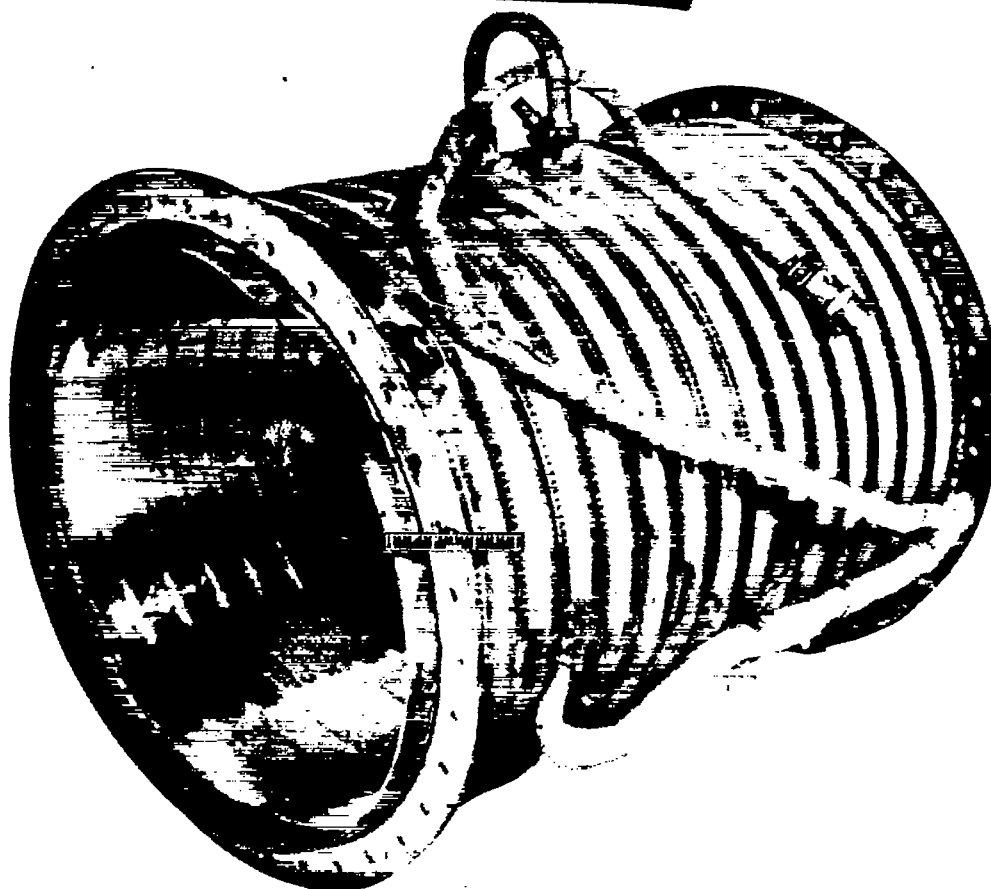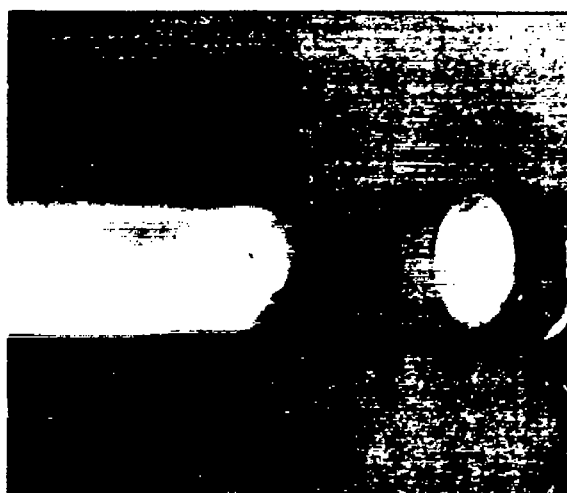of 20-inch ram jet.

684.

Figure 4. - Modified combustion-chamber section used in wind-
tunnel investigation of 20-inch ram jet.

NACA
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Figure 5. - Exhaust jet issuing at supersonic speeds from
exhaust nozzle of 20-inch ram jet during wind-tunnel
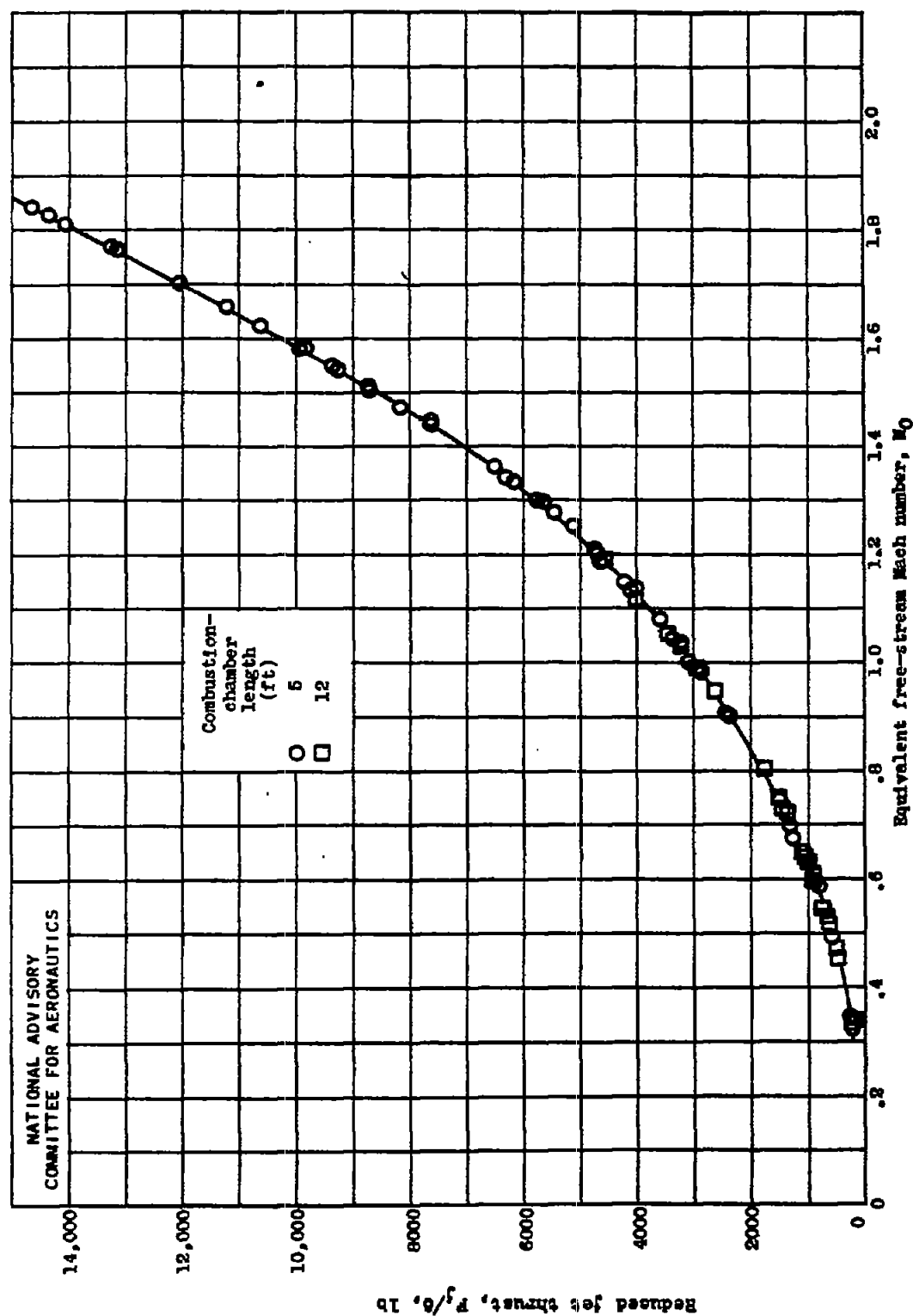investigation.

Figure 6.- Effect of equivalent free-stream Mach number M_0 and combustion-chamber length on reduced jet thrust F_j/δ. 20-inch ram jet with 16.8-inch-diameter exhaust nozzle; jet thrust reduced to NACA standard atmospheric conditions at sea level.
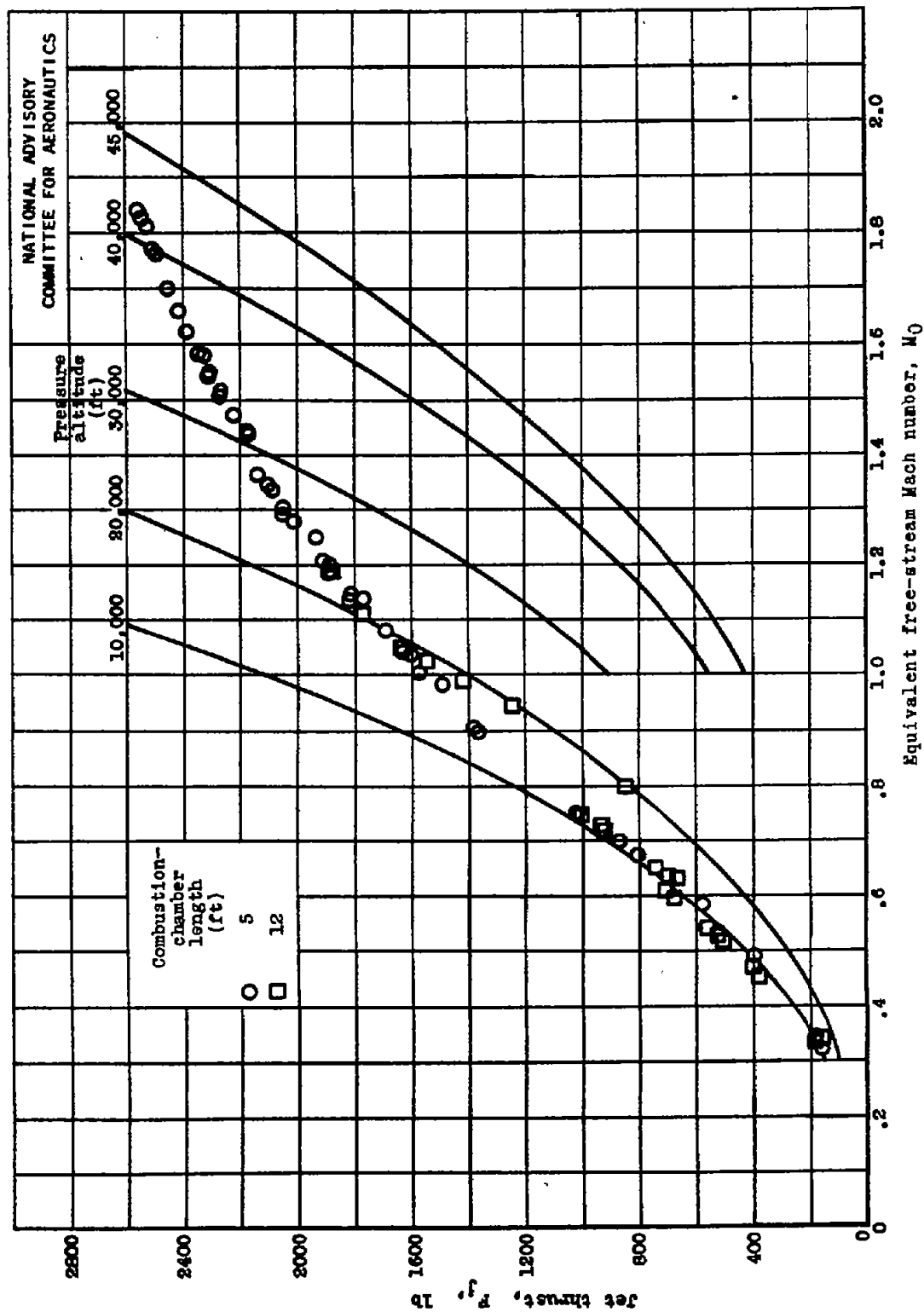
Fig. 7

Figure 7.— Effect of equivalent free-stream Mach number M_0, combustion-chamber length, and pressure altitude on jet thrust F_j. 20-inch ram jet with 16.8-inch-diameter exhaust nozzle.
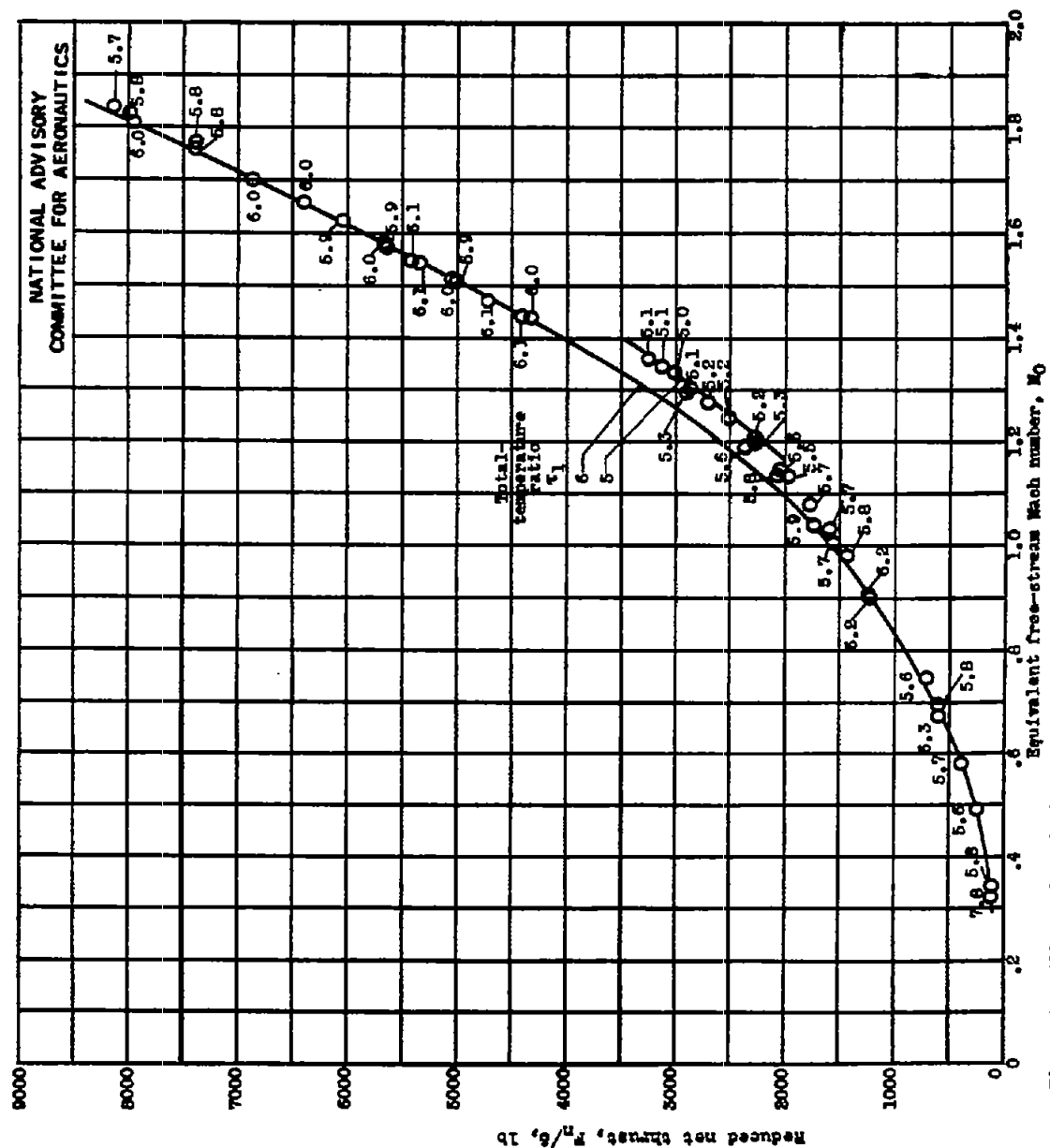
Fig. 8

Figure 8.– Effect of equivalent free-stream Mach number M_0 and total-temperature ratio τ_1 on reduced net thrust F_n/δ. 20-inch ram jet with 5-foot combustion chamber and 16.8-inch-diameter exhaust nozzle; net thrust reduced to NACA standard atmospheric conditions at sea level.

Fig. 9   CONFIDENTIAL   NACA RM No. E6L06
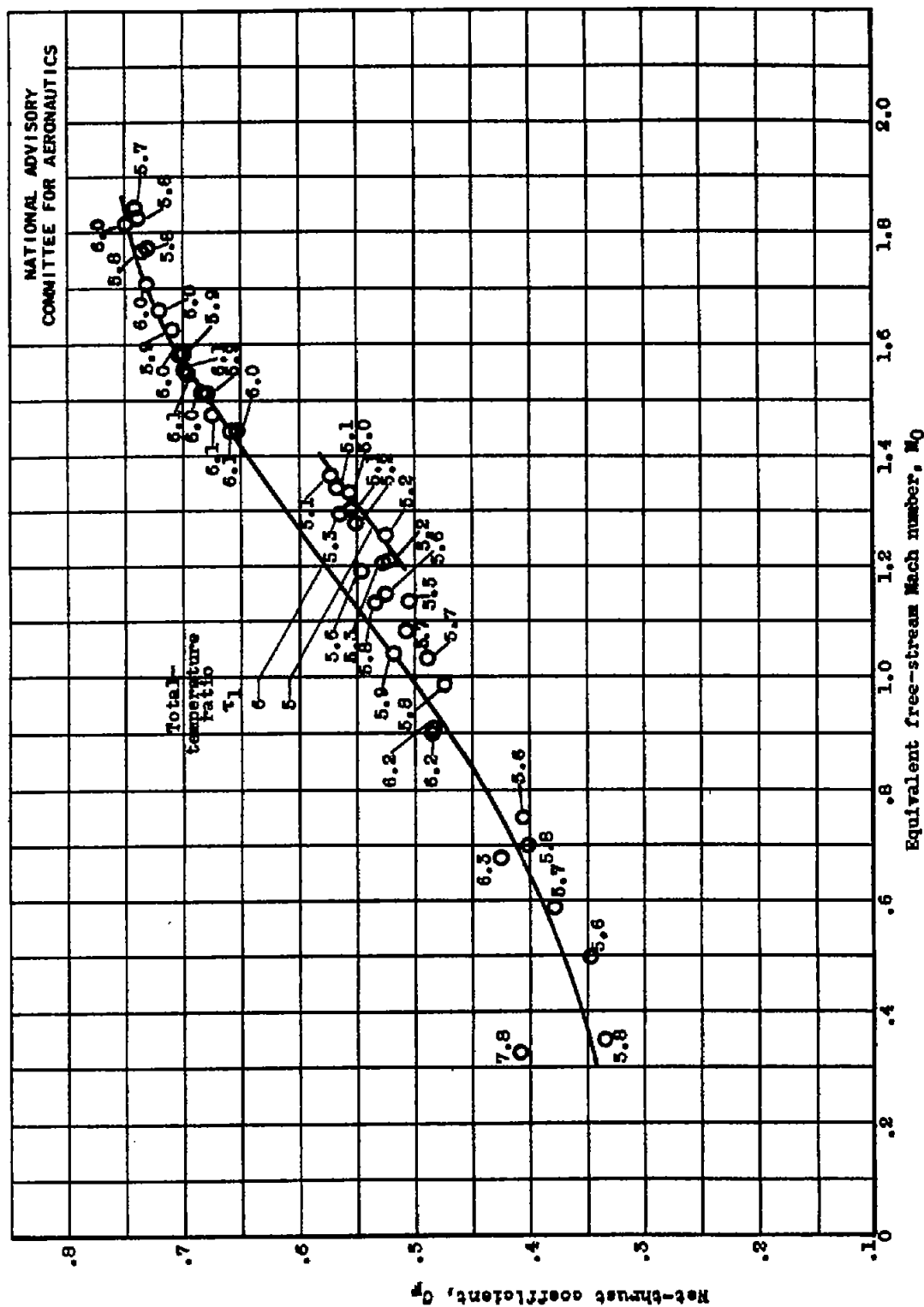
Figure 9.- Effect of equivalent free-stream Mach number M_0 and total-temperature ratio τ_1 on net-thrust coefficient C_F. 20-inch ram jet with 5-foot combustion chamber and 16.8-inch-diameter exhaust nozzle.

Figure 10.- Effect of equivalent free-stream Mach number M_0 on ideal over-all efficiency parameter $\frac{L}{\eta_b}$ $(1+\sqrt{\tau_1})$. 20-inch ram jet with 5-foot combustion chamber and 16.8-inch-diameter exhaust nozzle.
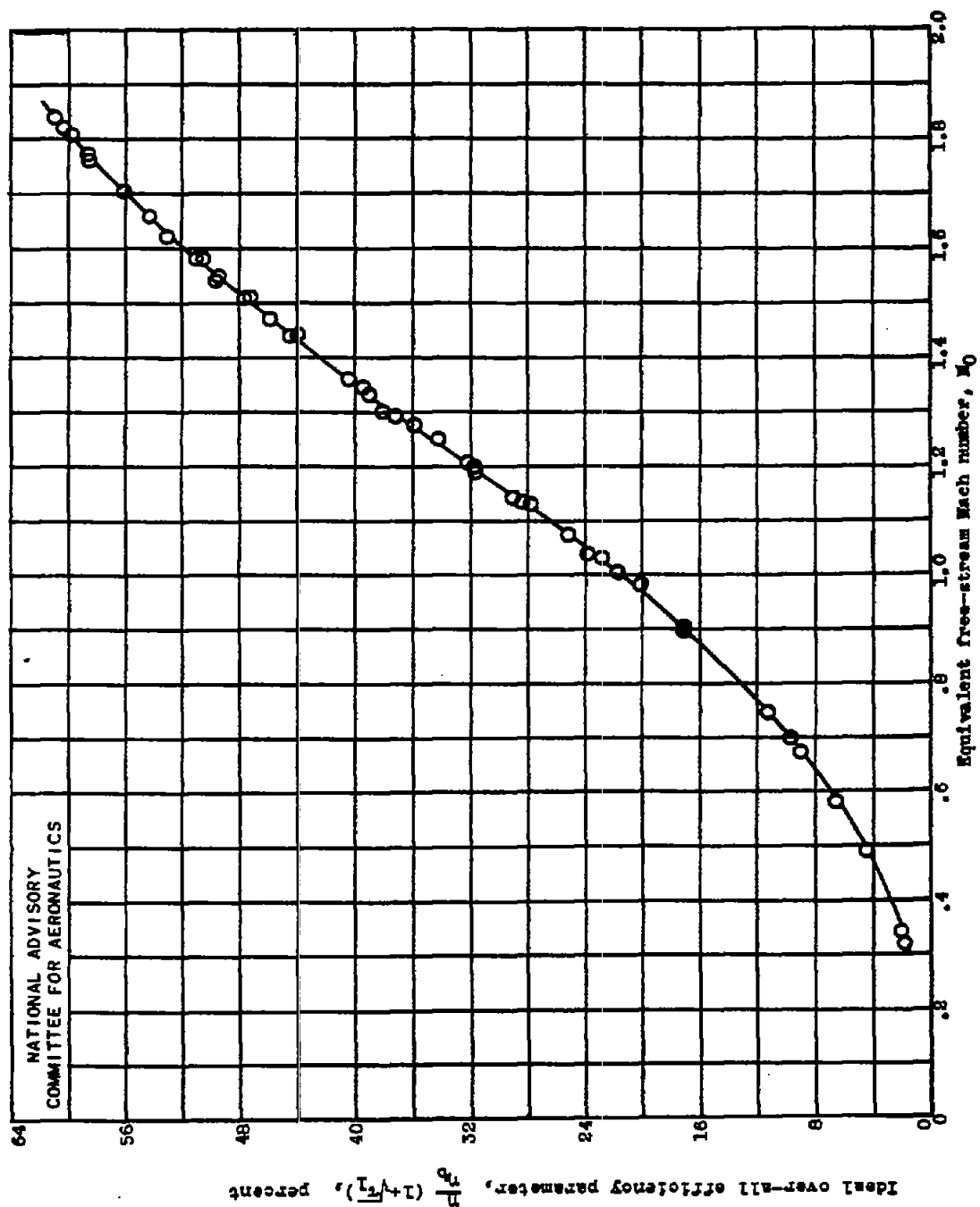
Figure 11.- Effect of equivalent free-stream Mach number M_0 on ideal net-power specific fuel consumption parameter $\dfrac{550\, W_f\, 3600}{F_n V_0 (1+\sqrt{r_1})}\eta_b$. 20-inch ram jet with 5-foot combustion chamber and 16.8-inch-diameter exhaust nozzle.
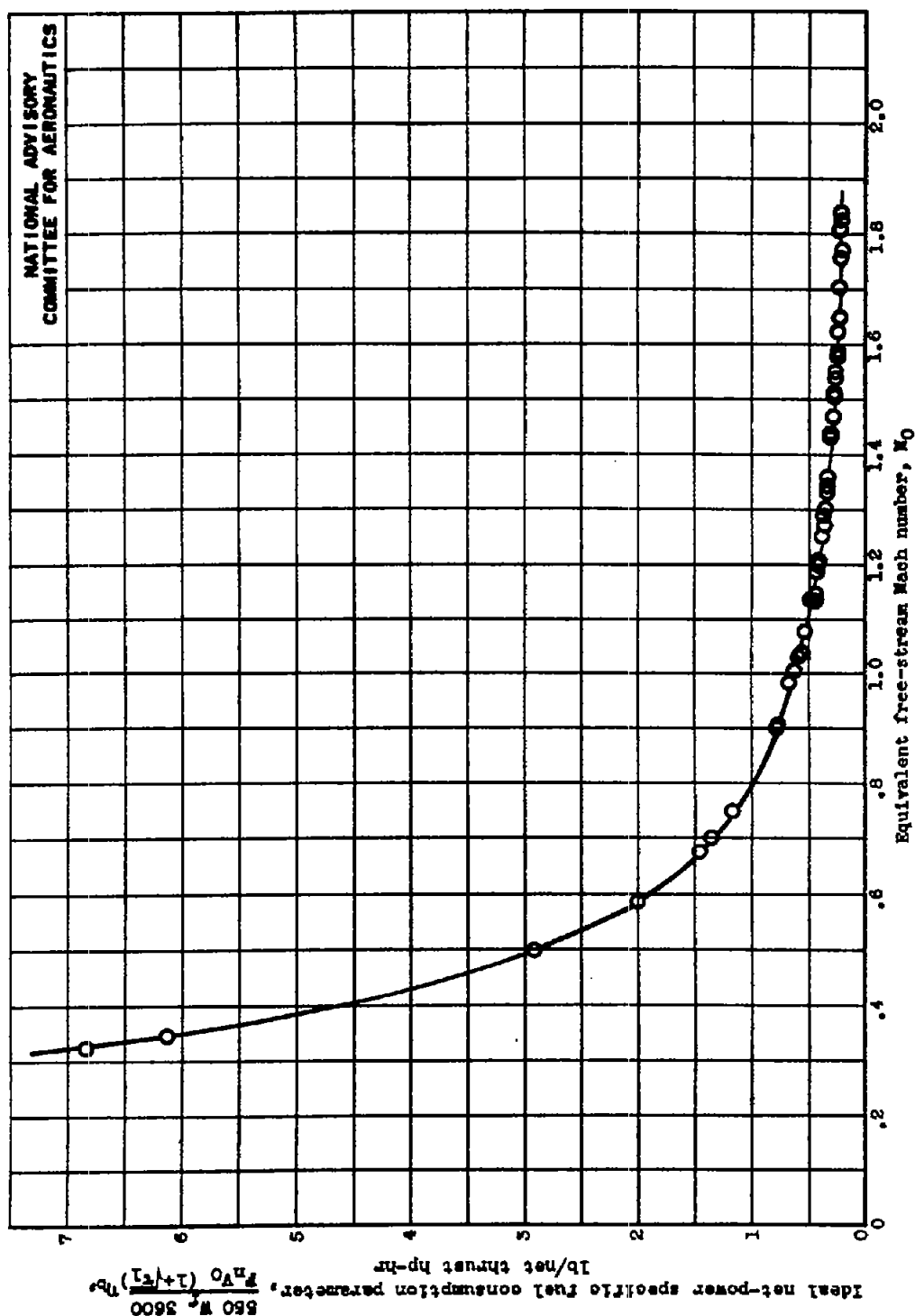
Equivalent free-stream Mach number, M_0

Ideal net-power specific fuel consumption parameter, $\dfrac{550\, W_f\, 3600}{F_n V_0 (1+\sqrt{r_1})}\eta_b$, lb/net thrust hp-hr

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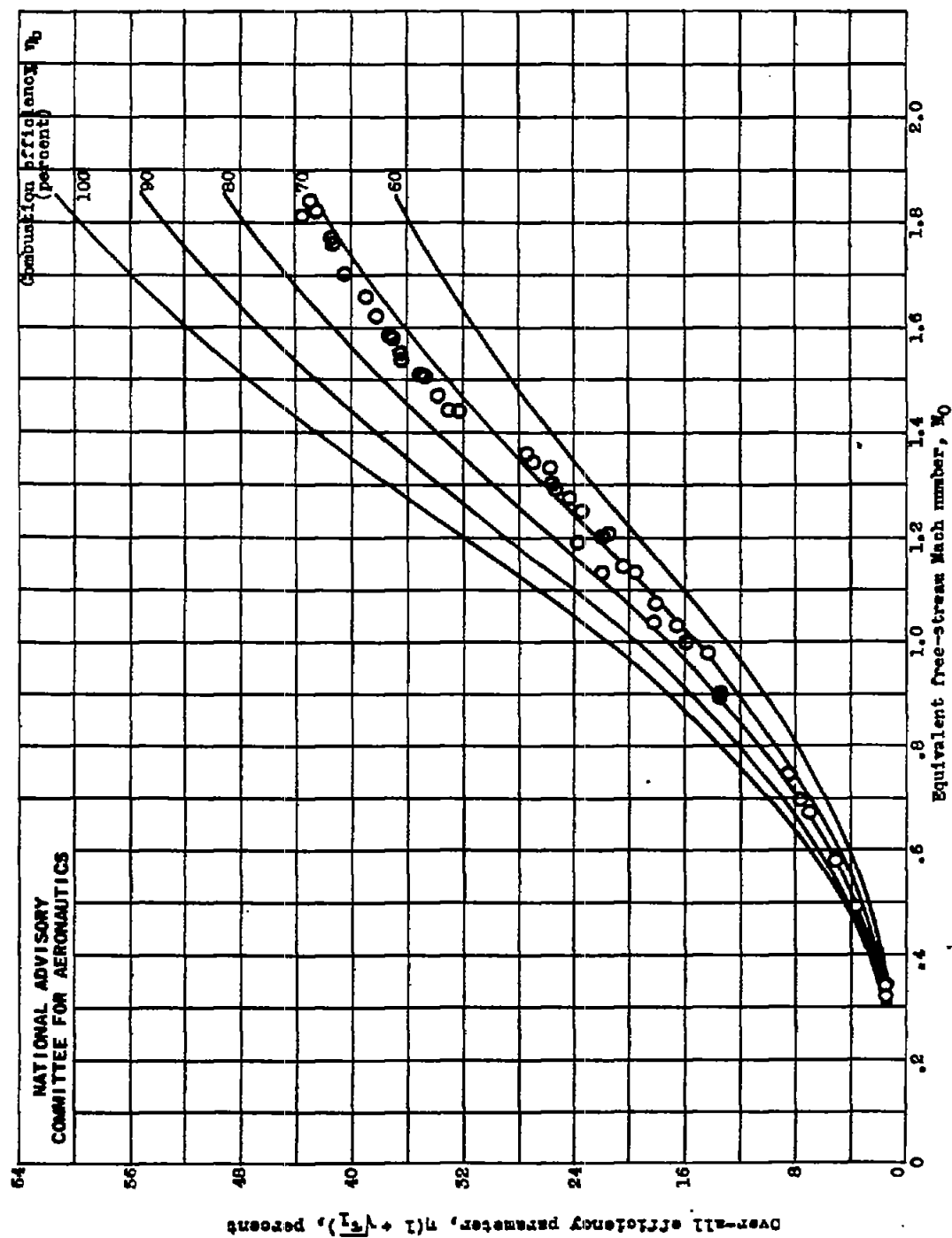
Fig. 12

Figure 12.— Effect of equivalent free-stream Mach number M_0 and combustion efficiency η_b on over-all efficiency parameter $\eta(1 + \sqrt{\tau_1})$. 20-inch ram jet with 5-foot combustion chamber and 16.8-inch-diameter exhaust nozzle.

Fig. 13   CONFIDENTIAL   NACA RM No. E6L06

Figure 13.— Effect of equivalent free-stream Mach number M_0 and combustion efficiency η_b on net-power specific fuel consumption parameter $\dfrac{550\ W_f\ 3600}{F_n V_0\ (14\sqrt{5_1})}$. 20-inch ram jet with 5-foot combustion chamber and 16.8-inch-diameter exhaust nozzle.
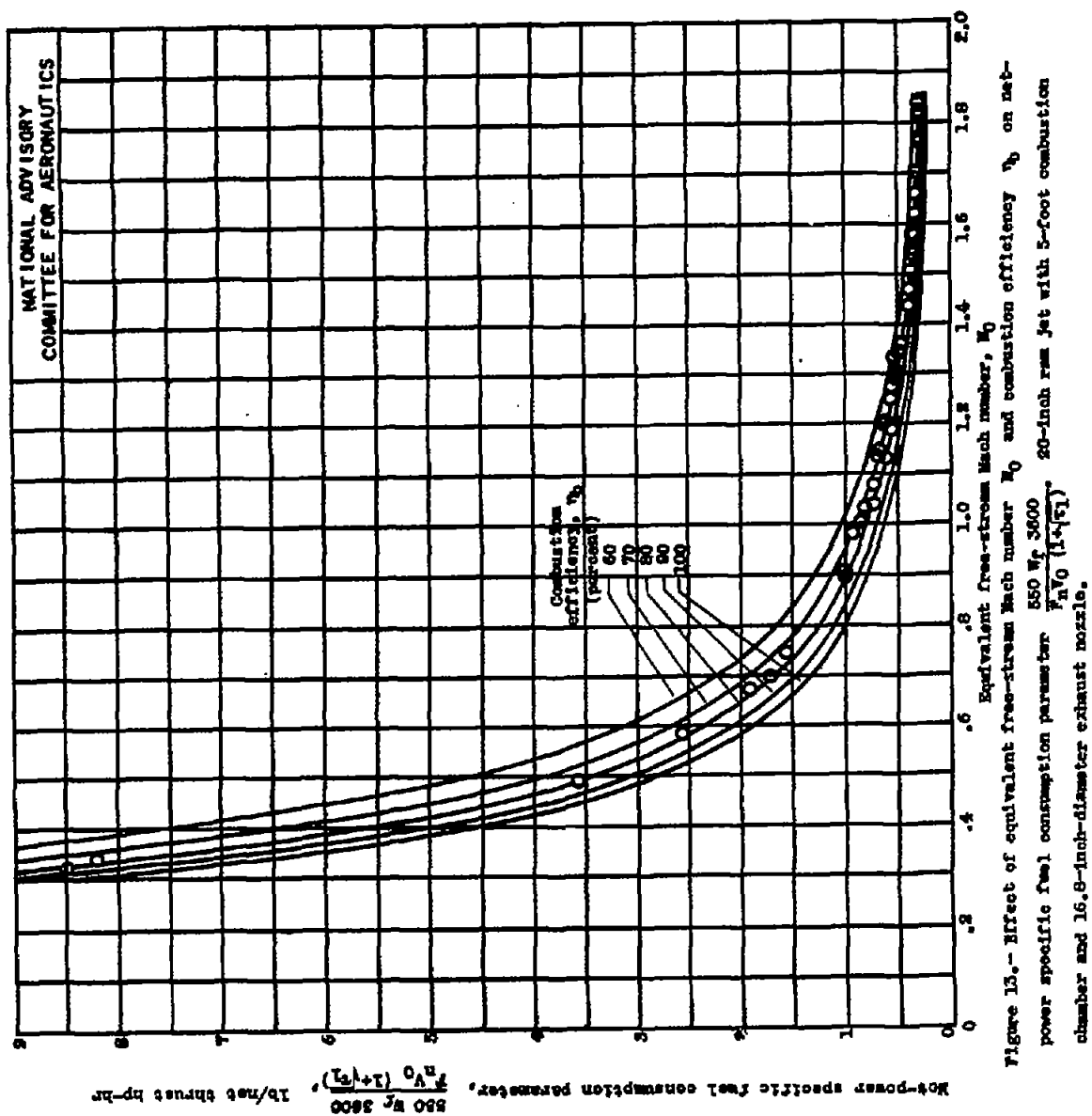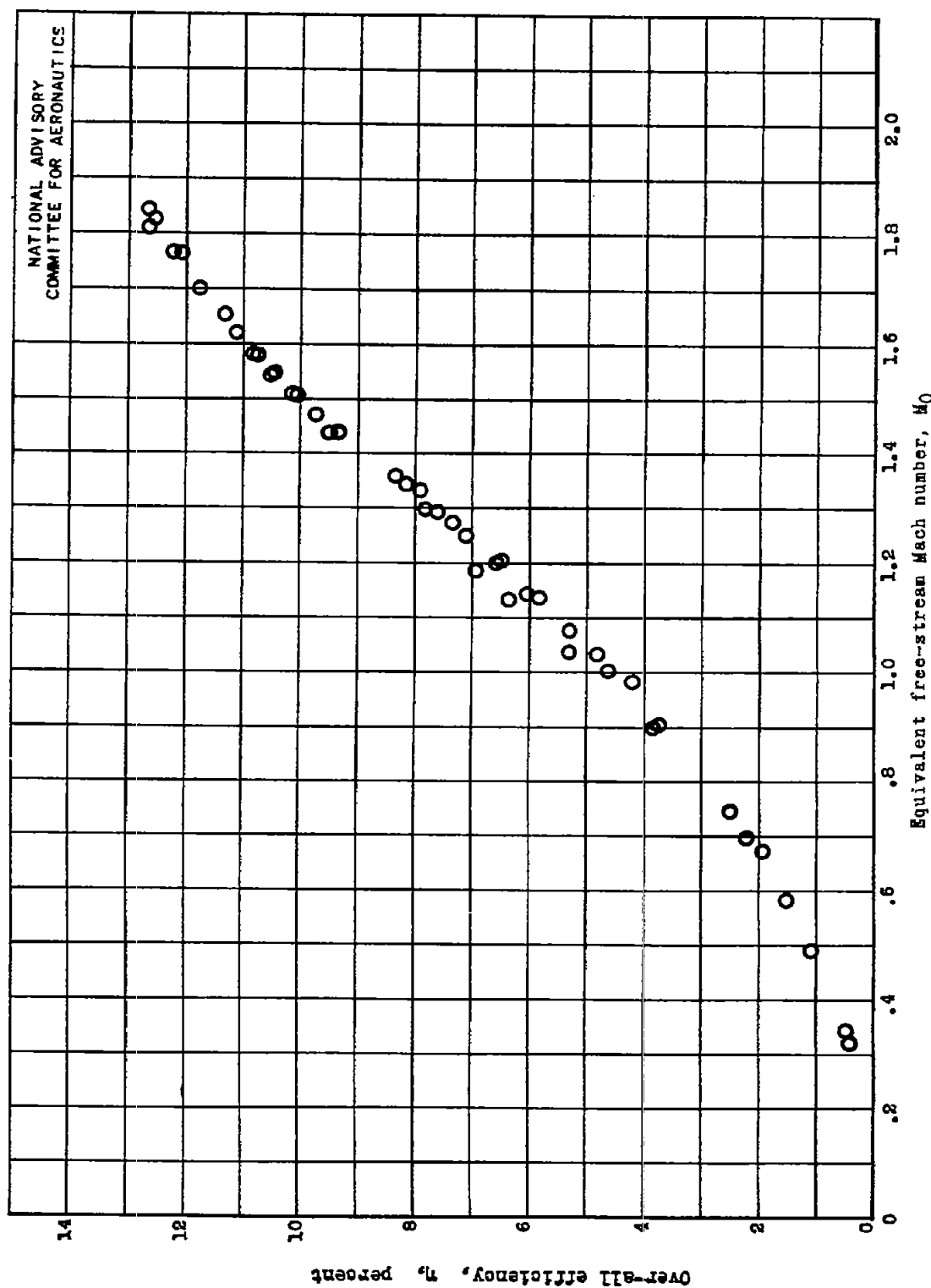
Figure 14.— Effect of equivalent free-stream Mach number M_0 on over-all efficiency η. 20-inch ram jet with 5-foot combustion chamber and 16.8-inch-diameter exhaust nozzle.

Fig. 15   CONFIDENTIAL   NACA RM No. E6L06
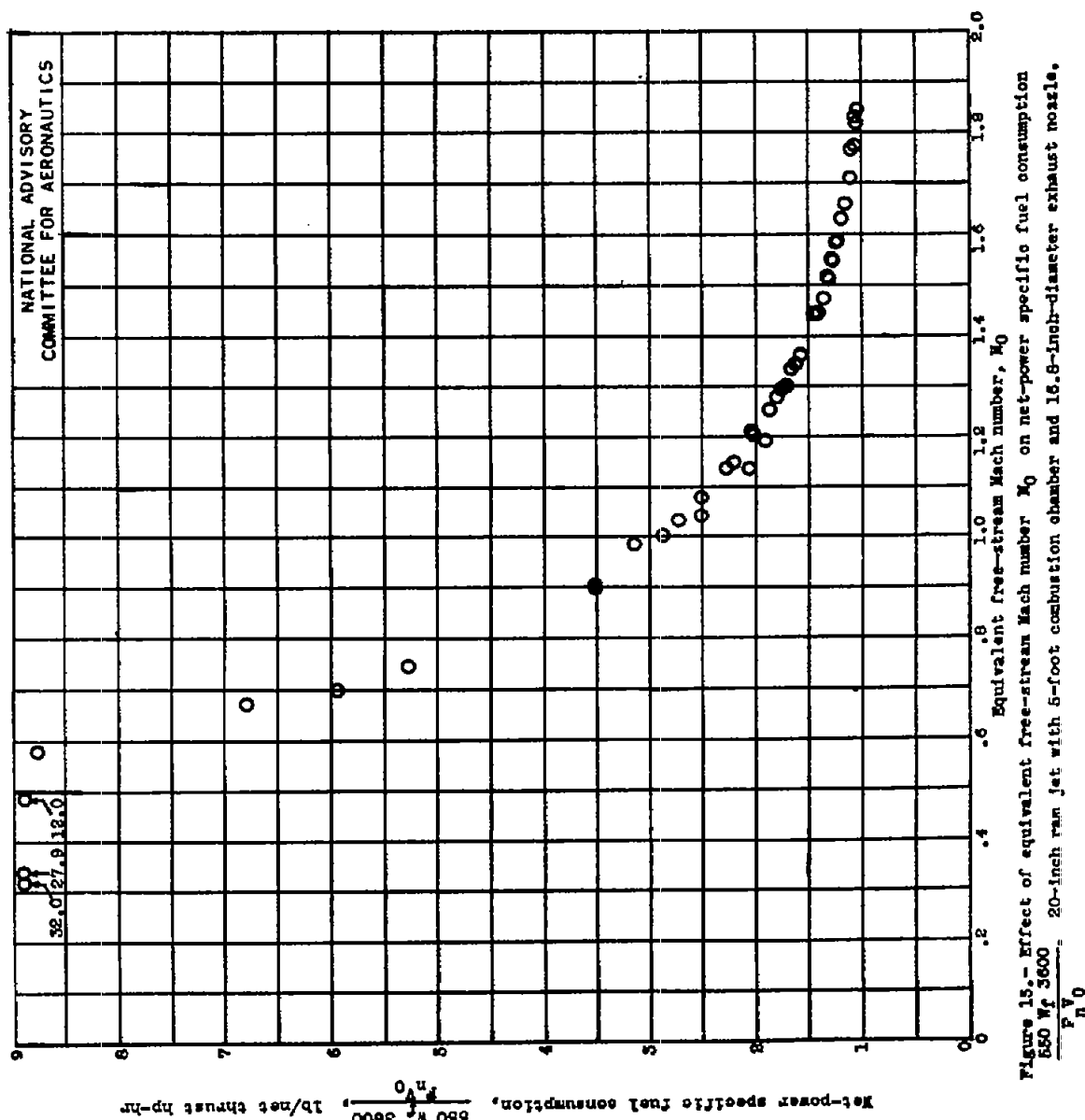

Figure 15.- Effect of equivalent free-stream Mach number M_0 on net-power specific fuel consumption $\dfrac{550\ W_f\ 3600}{F_n V_0}$. 20-inch ram jet with 5-foot combustion chamber and 15.8-inch-diameter exhaust nozzle.

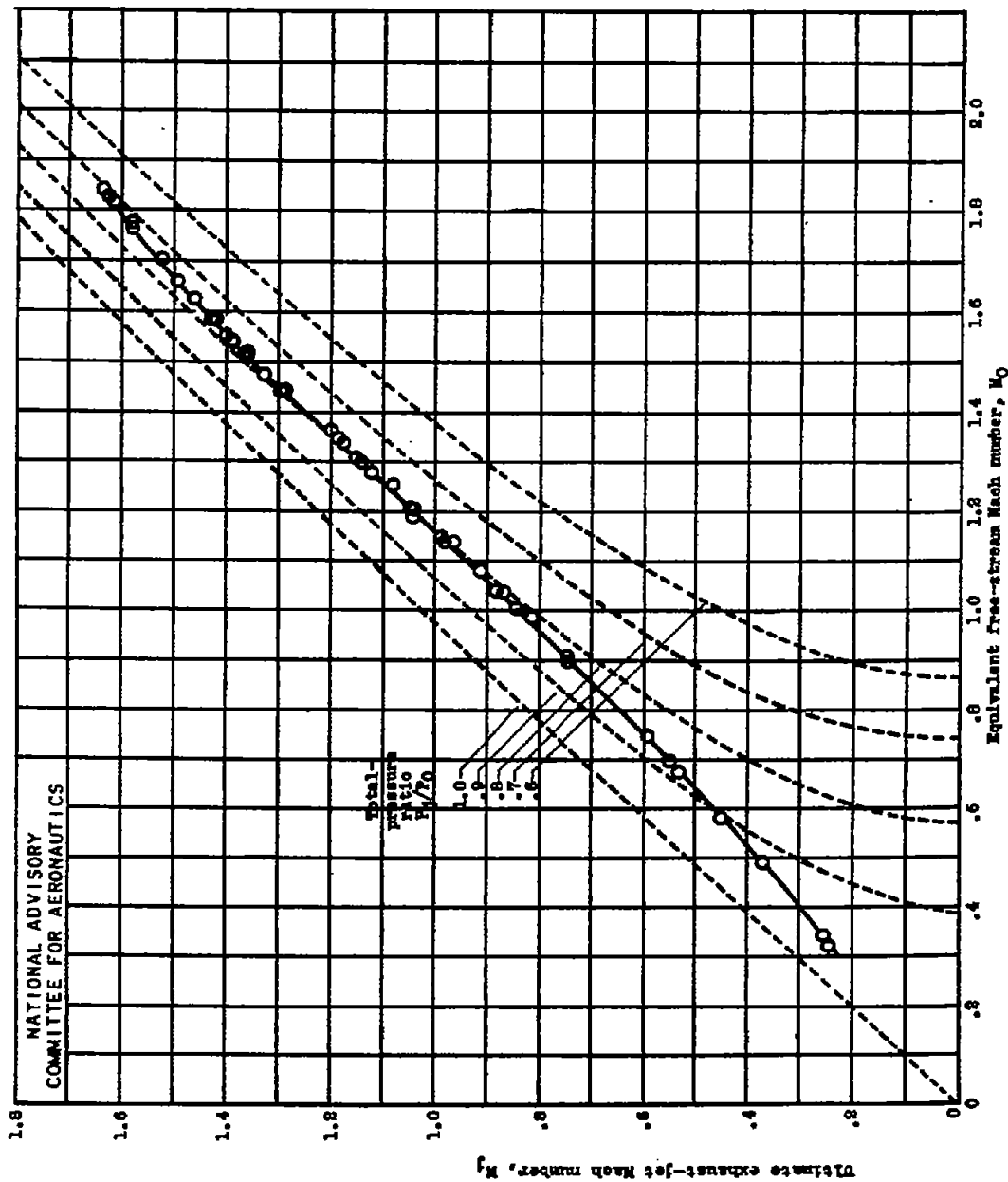Figure 16.- Effect of equivalent free-stream Mach number M_0 and total-pressure ratio across ram jet P_1/P_0 on ultimate exhaust-jet Mach number M_j. Total-pressure ratio curves calculated for ratio of specific heats at exhaust-jet condition $\gamma_1 = 1.3$ and at free-stream condition $\gamma_0 = 1.4$. 20-inch ram jet with 5-foot combustion chamber and 18.5-inch-diameter exhaust nozzle.

Fig. 17   NACA RM No. E6L06

NATIONAL ADVISORY
COMMITTEE FOR AERONAUTICS

Combustion-chamber-inlet Mach number parameter, $M_2\sqrt{\tau_2}$
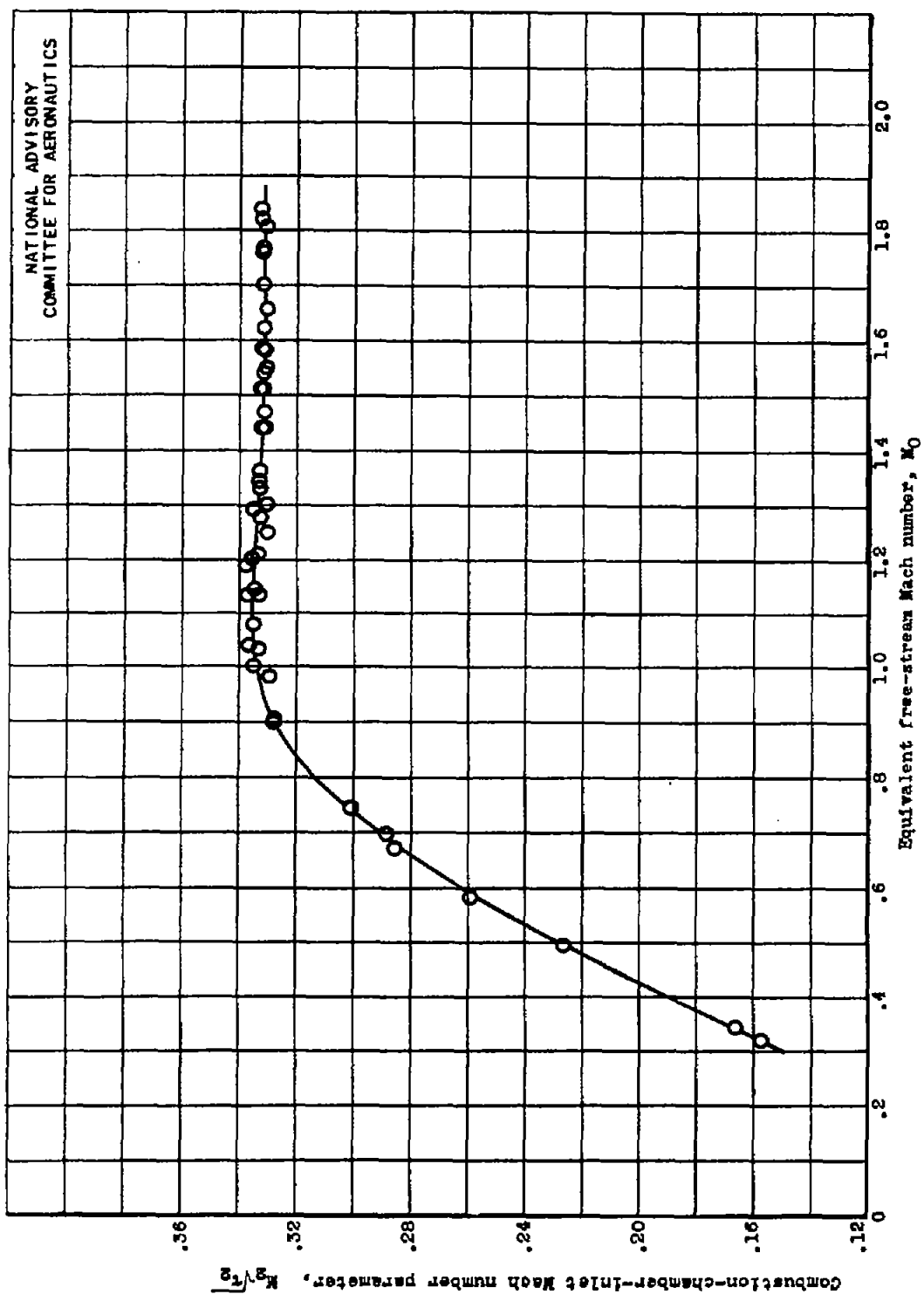
Equivalent free-stream Mach number, M_0

Figure 17.- Effect of equivalent free-stream Mach number M_0 on combustion-chamber-inlet Mach number parameter $M_2\sqrt{\tau_2}$. 20-inch ram jet with 5-foot combustion chamber and 16.8-inch-diameter exhaust nozzle.
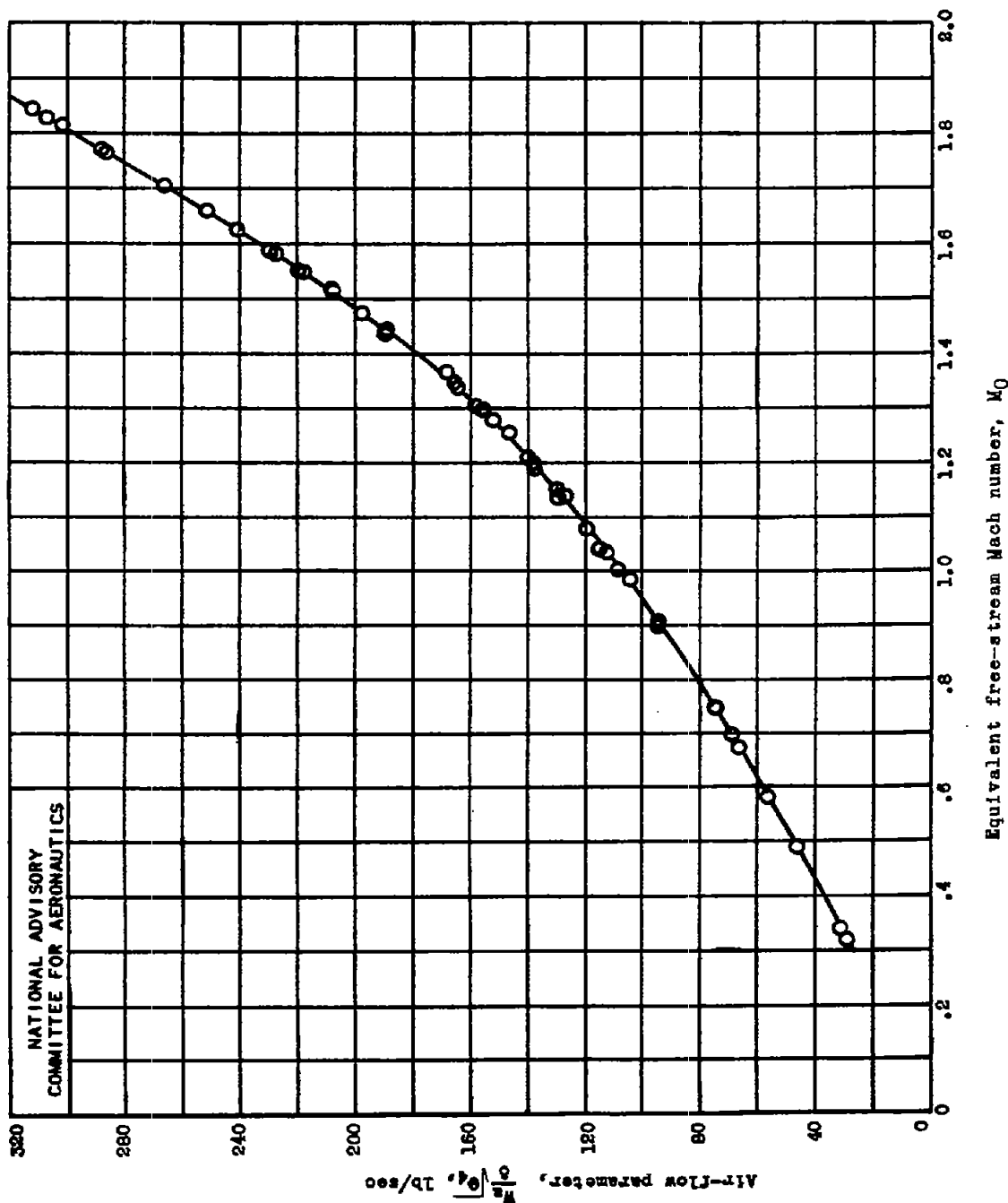
Fig. 18

Figure 18.-Effect of equivalent free-stream Mach number M_0 on air-flow parameter $\frac{W_a}{\delta}\sqrt{\theta_4}$. 20-inch ram jet with 5-foot combustion chamber and 16.8-inch-diameter exhaust nozzle.
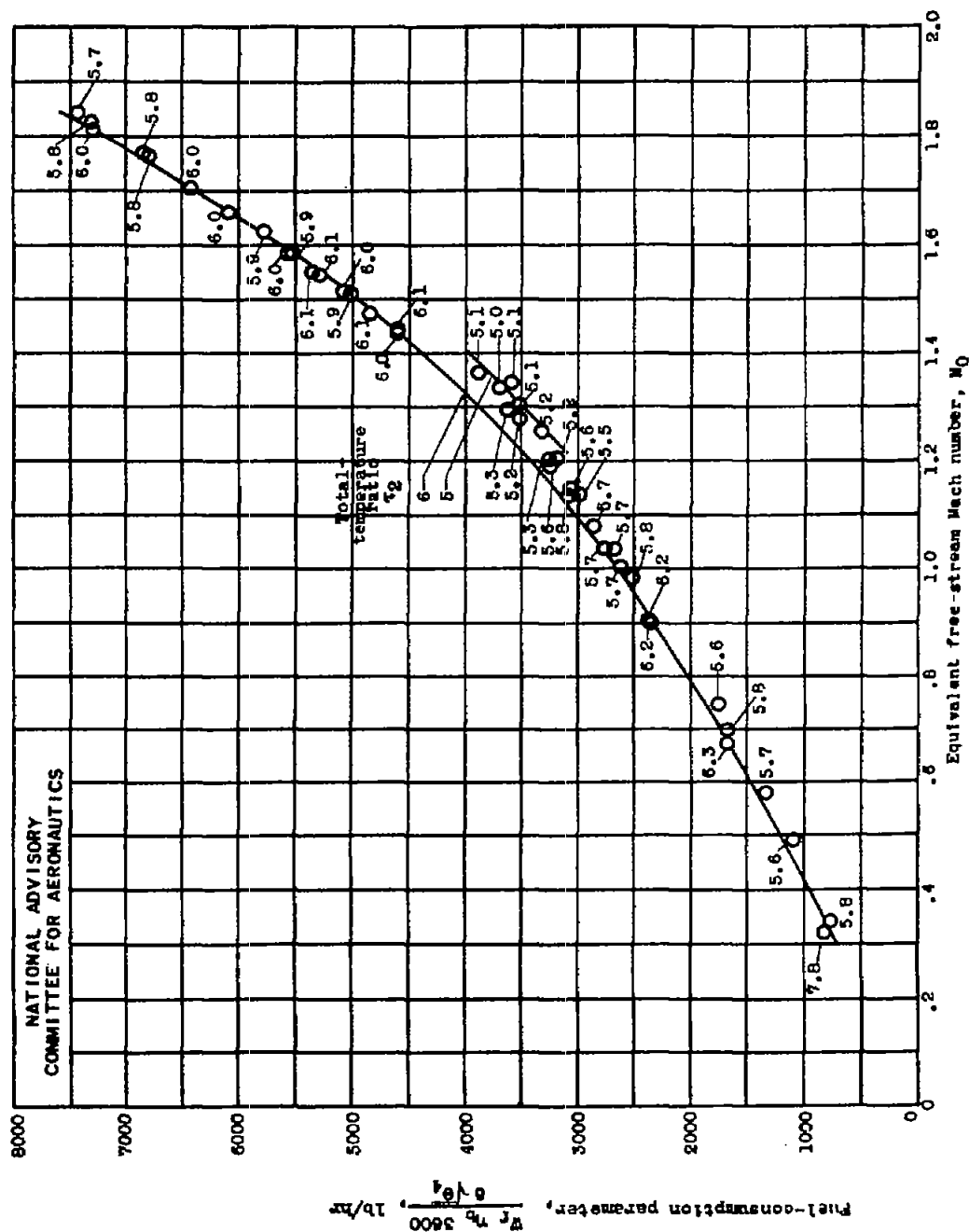
Fig. 19　　　　CONFIDENTIAL　　　　NACA RM No. E6L06

Figure 19.- Effect of equivalent free-stream Mach number M_0 and total-temperature ratio τ_2 on fuel-consumption parameter $\dfrac{W_f}{\eta_b} \dfrac{3600}{\theta_1 \sqrt{\theta_4}}$. 20-inch ram jet with 5-foot combustion chamber and 16.8-inch-diameter exhaust nozzle.

Fig. 20

Figure 20.- Effect of fuel-air ratio f/a, combustion-chamber-inlet absolute static pressure p_2 and Mach number M_2, and combustion-chamber-exit absolute static pressure (tunnel ambient pressure) p_0 on combustion efficiency η_b. 20-inch ram jet with 5-foot combustion chamber and 16.8-inch diameter exhaust nozzle.

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Combustion efficiency, η_b, percent

Fuel-air ratio, f/a

1722 1264 .1002

1977 1289 .940

1952 1301 1000

1938 1125 .1403

1820 1025 1400

1486 1166 1000

1895 0900 1602

2001 1025 1600

1714 .0640 1602

1622 0903 1602

1919 0953 1600

1968 1183 1400

1819 1288 1000

1903 1333 1002

1979 1323 660

1992 1082 1603

1928 1182 1400

1787 1125 1400

1903 0899 1602

1852 0916 1600

1796 1106 1400

1720 Combustion-chamber-inlet static pressure
0719 Combustion-chamber-inlet Mach number M_2
1600 Exhaust-nozzle-exit static pressure p_0

f/a, combustion-chamber-inlet absolute static
pressure P_2, and combustion-chamber-exit absolute static
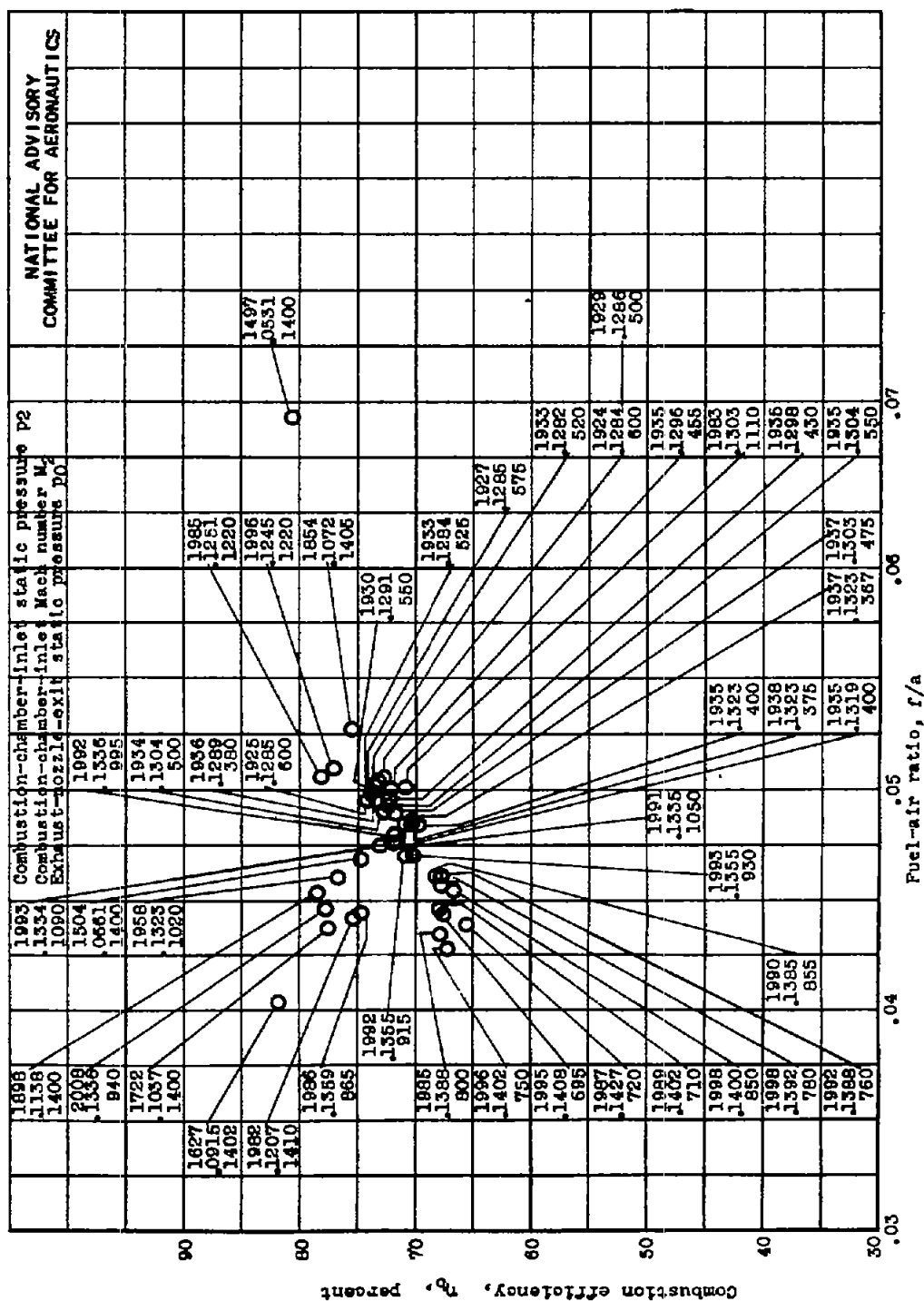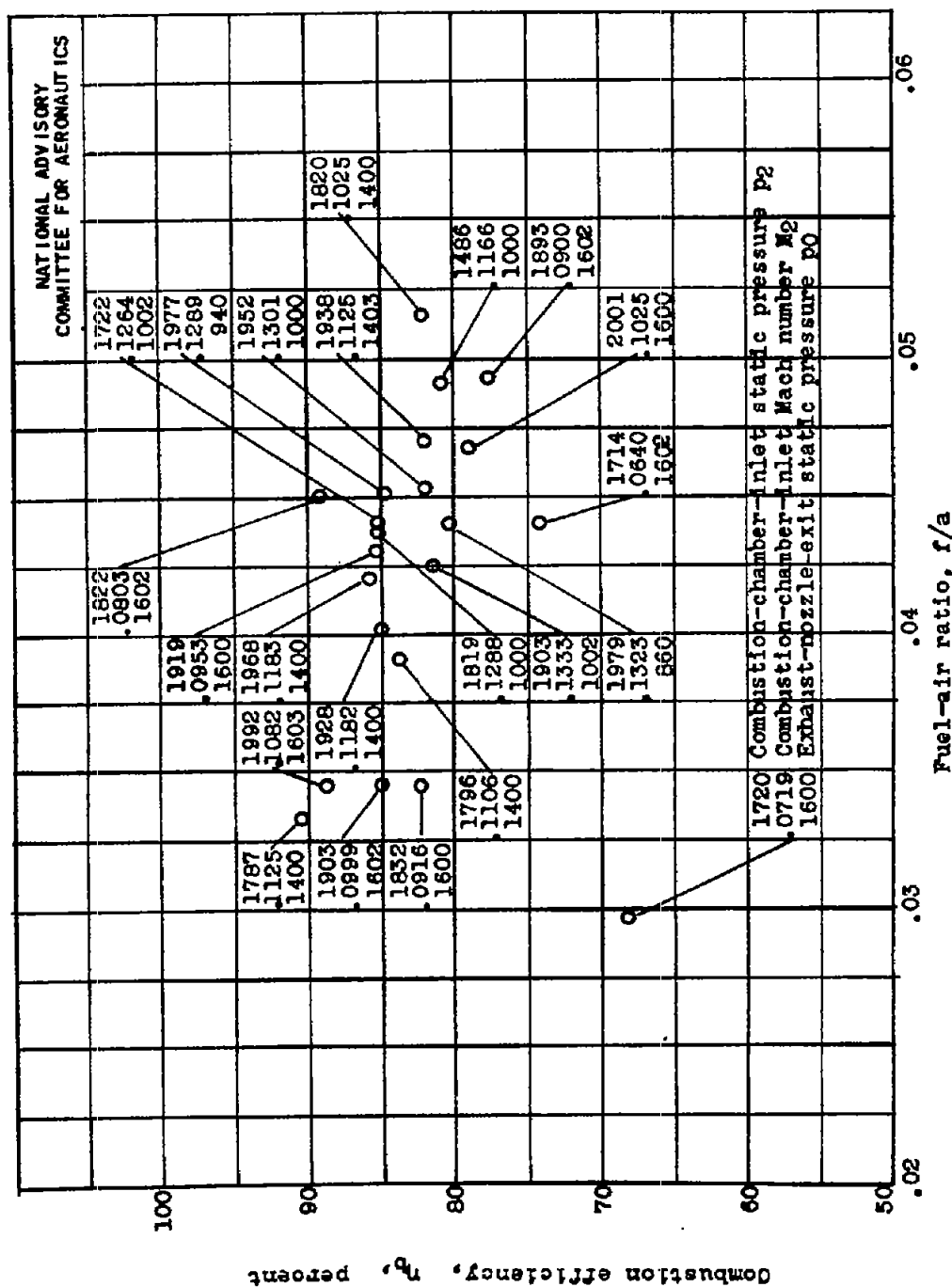pressure P_0 on combustion efficiency η_b.

Figure 21.- Effect of fuel-air ratio f/a, combustion-chamber-inlet absolute static
pressure P_2 and Mach number M_2, and combustion-chamber-exit absolute static
pressure (tunnel ambient pressure) p_0 on combustion efficiency η_b. 20-inch ram
jet with 12-foot combustion chamber and 16.8-inch diameter exhaust nozzle.

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ABSTRACT

Investigation was conducted on performance of 20-in. ramjet engine over wide range of altitudes and free-stream Mach numbers. At maximum free-stream Mach number 1.84, net thrust of 8135 lb and over-all efficiency of 12.6% were attained. Specific fuel consumption was 1.06 lb per net thrust horsepower-hour, and combustion efficiency was 70.3%. Principal total pressure loss was caused by combustion. Method of reducing data was improved to include effects of variables at high Mach numbers. No performance variation occurred when fuel was changed from preheated 62 octane to preheated 100 octane gasoline.

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